NAVAL SHIP RESEARCH AND DEVELOPMENT CENTER
Bethesda, Maryland 20034

CALCULATION OF THE HEAT TRANSFER ON THE WINDWARD SIDE OF A DELTA WING IN HYPERSONIC FLOW

by
Stephen Sacks

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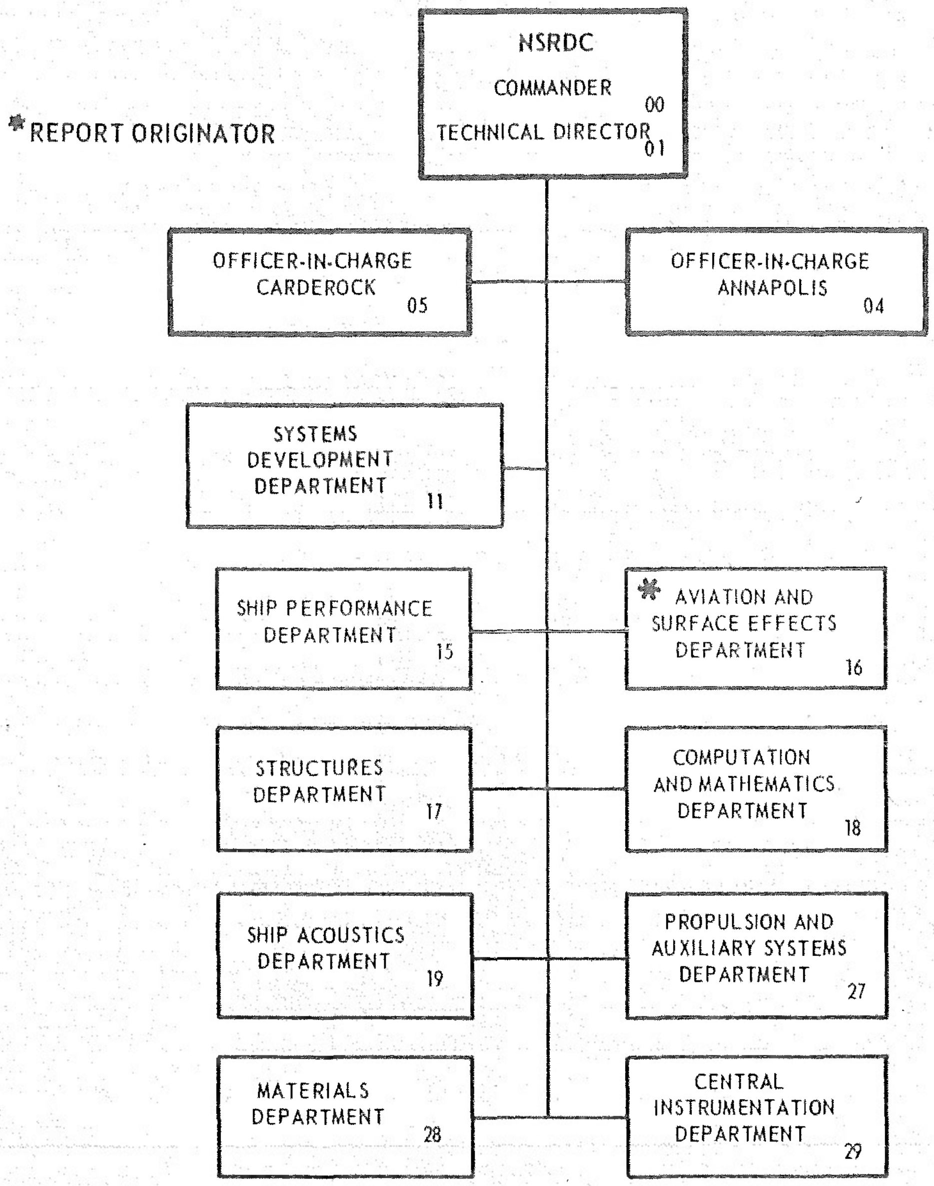

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NAVAL SHIP RESEARCH AND DEVELOPMENT CENTER

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CALCULATION OF THE HEAT TRANSFER ON THE WINDWARD SIDE OF A DELTA WING IN HYPERSONIC FLOW

1. Triangular wing -- Heat transfer

2. " " -- Hypersonic flow

by

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NOTATION

θ	Angle between the tangent to a local streamline and the radial line
Λ	Wing sweep angle
μ	Viscosity
ν	Equals μ/p
ρ	Density
γ	Ratio of specific heats
ϕ	Angular coordinate defined separately on sphere, cylinder, and flat plate elements
C_p	Specific heat at constant pressure
H	Enthalpy
h_1,h_2,h_3	Length elements for the curvilinear coordinate system
M	Mach number
P	Pressure
Pr	Prandtl number
q	Heat Transfer rate
R_o	Radius of spherical nose on the delta wing
R_G	Gas constant
S,s	Streamline length
T	Temperature
u	Velocity
V_1,V_2,V_3	Velocity components along the curvilinear coordinate directions
x_1,x_2,x_3	Orthogonal curvilinear coordinates
x	Length coordinate defined separately on sphere, cylinder, and flat plate elements

SUBSCRIPTS

aw	Adiabatic wall
b	Cylinder values
c	Flat plate values
e	At the boundary layer edge

f	Final value
i	Initial value
o	Stagnation value behind nose shock wave
oo	Upstream infinity stagnation value
r	At the reference condition
w	At the wall
∞	At upstream infinity

ABSTRACT

A computer program has been written to calculate the heat transfer on the windward side of a delta wing in hypersonic flow. The program utilizes the small crossflow theory of Vaglio-Laurin. Heat transfer values are calculated for both the laminar and turbulent boundary layer cases, and results are compared with experimental data on delta wings available in the literature.

ADMINISTRATIVE INFORMATION

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INTRODUCTION

The objective of this work was to develop a computer program to predict the heat transfer characteristics on the windward side of a delta wing at hypersonic speeds. The program could then be used to determine the thermal environment of the configuration.

The method of approach was based on the work of Vaglio-Laurin,[1,2*] who demonstrated that on three-dimensional bodies with highly cooled surfaces and moderate Mach numbers at the outer edge of the boundary layer, crossflow terms in the boundary layer were negligible--even for large transverse pressure gradients. With this assumption, heat transfer characteristics on the three-dimensional body could be found using formulas derived for axisymmetric (no crossflow) bodies provided that the pressure distribution on the three-dimensional body be known.

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[1]Vaglio-Laurin, R., "Laminar Heat Transfer on Three-Dimensional Blunt Nosed Bodies in Hypersonic Flow," America Rocket Society Journal New York, p. 123-129 (Feb 1959).

[2]Vaglio-Laurin, R., "Turbulent Heat Transfer on Blunt-Nosed Bodies in Two-Dimensional and General Three-Dimensional Hypersonic Flow," Journal Aero/Space Science New York, p. 27-36 (Jan 1960).

[*]A complete list of References is listed on page 35.

The calculation is in two parts. First, the inviscid streamline pattern and thermodynamic property distribution in streamline coordinates are found on the body surface. Second, at each calculated point on the body surface, analogy is made with various axisymmetric heat transfer equations, and the heat transfer characteristics are determined. The method of Lees[3] and Vaglio-Laurin[1] are used for laminar flow; a streamline divergence turbulence flow method[4] and a modified method by Rose, Probstein, and Adams[5] are used for turbulent flow.

The actual calculation begins from the stagnation point on the spherical nose, proceeds to the cylindrical leading edge, and goes on to the flat plate regions. Appropriate integrals of thermodynamic properties are calculated along the streamlines to permit heat transfer analysis. Various pressure distributions are used in each region. The boundary layer entropy values used herein were determined from the normal shock at the nose.

The report presents an analytic development of the small crossflow approximation as derived by Vaglio-Laurin, gives the specific governing equations for the streamline pattern on a delta wing, and presents the pressure distribution expressions that apply to this geometry. Then the thermodynamic variables required in the analysis are discussed along with the governing heat transfer equations that use these variables. Results obtained by using this theory are compared with experimental results available in the literature for blunt-nosed hypersonic layers. These comparsions include calculated and experimentally determined pressures, resulting streamline patterns, and computed and measured heat transfer values.

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[3] Lees, L., "Laminar Heat Transfer over Blunt-Nosed Bodies at Hypersonic Flight Speeds," America Rocket Society Journal New York, p. 259-269 (Apr 1956).

[4] Ratliff, A.W. et al., "Analysis of Heating Rates and Forces on Bodies Subject to Rocket Exhaust Plume Impingement," Huntsville, Ala., Lockheed Missiles & Space Co., LM SC/HREC A79 1230, HREC 1150-1, Contract NAS8-21150 (Mar 1968).

[5] Tai, T.C., "Laminar and Turbulent Convective Heat Transfer over Bodies at an Angle of Attack." Virginia Polytechnic Institute, Thesis, Blacksburg, Va. (Oct 1968).

ANALYTICAL DEVELOPMENT

The equations for the steady, laminar boundary layer flow of a homogeneous gas in the absence of body forces and heat sources may be written in an orthogonal (x_1, x_2, x_3) coordinate system as (in this section only h_L refers to local enthalpy and H to stagnation enthalpy)

$$\left(\frac{\partial}{\partial x_1}\right)\left[h_2 h_3 \rho V_1\right] + \left(\frac{\partial}{\partial x_2}\right)\left[h_1 h_3 \rho V_2\right] + \left(\frac{\partial}{\partial x_3}\right)\left[h_1 h_2 \rho V_3\right] = 0 \tag{1a}$$

$$V_1\left(\frac{\partial V_1}{h_1 \partial x_1}\right) + V_2\left(\frac{\partial V_1}{h_2 \partial x_2}\right) + V_3\left(\frac{\partial V_1}{h_3 \partial x_3}\right) + \left(\frac{V_1 V_2}{h_1 h_2}\right)\left(\frac{\partial h_1}{\partial x_2}\right)$$
$$-\left(\frac{V_2^{\,2}}{h_1 h_2}\right)\left(\frac{\partial h_2}{\partial x_1}\right) = -\left(\frac{1}{\rho}\right)\left(\frac{\partial P}{h_1 \partial x_1}\right) + \left(\frac{1}{\rho}\right)\left(\frac{\partial}{h_3 \partial x_3}\right)\left[\mu\left(\frac{\partial V_1}{h_3 \partial x_3}\right)\right] \tag{1b}$$

$$V_1\left(\frac{\partial V_2}{h_1 \partial x_1}\right) + V_2\left(\frac{\partial V_2}{h_2 \partial x_2}\right) + V_3\left(\frac{\partial V_2}{h_3 \partial x_3}\right) + \left(\frac{V_1 V_2}{h_1 h_2}\right)\left(\frac{\partial h_2}{\partial h_1}\right)$$
$$-\left(\frac{V_1^{\,2}}{h_1 h_2}\right)\left(\frac{\partial h_1}{\partial x_2}\right) = -\left(\frac{1}{\rho}\right)\left(\frac{\partial P}{h_2 \partial x_2}\right) + \left(\frac{1}{\rho}\right)\frac{\partial}{h_3 \partial x_3}\left[\mu\left(\frac{\partial V_2}{h_3 \partial x_3}\right)\right] \tag{1c}$$

$$P = P_e + 0\,[\delta] \tag{1d}$$

$$V_1\left(\frac{\partial H}{h_1 \partial x_1}\right) + V_2\left(\frac{\partial H}{h_2 \partial x_2}\right) + V_3\left(\frac{\partial H}{h_3 \partial x_3}\right)$$
$$= \left(\frac{1}{\rho}\right)\left(\frac{\partial}{h_3 \partial x_3}\right)\left[\mu\left\{\frac{\partial H}{h_3 \partial x_3} + \left(\frac{(1-P_r)}{P_r}\right)\left(\frac{\partial h_L}{h_3 \partial x_3}\right)\right\}\right] \tag{1e}$$

These results are taken directly from Vaglio-Laurin.[2] Choosing x_1 coincident with the inviscid-flow streamline at the outer edge of the boundary layer and x_3 normal to the surface, one has as boundary conditions

$$x_3 = 0 \qquad V_1 = V_2 = V_3 = 0\,; \qquad H = H_w \tag{2a}$$

$$x_3 \to \infty \qquad V_1 \to V_{1e}\,; \qquad V_2 \to 0\,; \qquad H = H_e \tag{2b}$$

For the case of highly cooled walls and $V_{1e}^2/(H_e - H_w) << 1$

$$\frac{V_1^2}{V_{1e}^2} - \frac{\rho_e}{\rho} \to 0 \tag{3}$$

As discussed in Reference 1, it can also be shown that under the subject conditions, $V_2 = 0$ (inside of the boundary layer). Noting that $h_3 = 1$, the governing equations become

$$\left(\frac{\partial}{\partial x_1}\right)\left(h_2 \rho V_1\right) + \left(\frac{\partial}{\partial x_3}\right)\left(h_2 \rho V_3\right) = 0 \tag{4a}$$

$$V_1 \left(\frac{\partial V_1}{\partial x_1}\right) + V_3 \left(\frac{\partial V_1}{\partial x_3}\right) = -\left(\frac{1}{\rho}\right)\left(\frac{\partial P}{\partial x_1}\right) + \left(\frac{1}{\rho}\right)\left(\frac{\partial}{\partial x_3}\right)\left[\mu \frac{\partial V_1}{\partial x_3}\right] \tag{4b}$$

$$P = P_e + 0\ [\delta] \tag{4c}$$

$$V_1 \left(\frac{\partial H}{\partial x_1}\right) + V_3 \left(\frac{\partial H}{\partial x_3}\right) = \left(\frac{1}{\rho}\right)\left(\frac{\partial}{\partial x_3}\right)\left[\mu \left\{\frac{\partial H}{\partial x_3} + \left[\frac{(1-P_r)}{P_r}\right]\frac{\partial h_L}{\partial x_3}\right\}\right] \tag{4d}$$

with the boundary conditions

$$x_3 = 0 \qquad V_1 = V_3 = 0\ ; \qquad H = H_w \tag{5a}$$

$$x_3 \to \infty \qquad V_1 = V_{1e} = u_e \qquad H = H_e \tag{5b}$$

Examination of the preceding equations shows that they become formally identical to those governing the boundary layer over an axisymmetric body at zero angle of attack, provided that h_2 be replaced by the radial coordinate r and distance be measured along respective streamlines. The same equations that govern the heat transfer on an axisymmetric body at zero angle of attack may also be used in the present nonaxisymmetric case, provided that r be replaced by h_2 in the axisymmetric heat transfer equations.

Three regions on the delta wing are under consideration: the spherical nose cap, the cylindrical leading edges, and the flat plate lower area.

Under the assumptions made in this study, all boundary layer streamlines go from the geometric stagnation point on the spherical nose, to the cylindrical leading edge, to the flat plate region. Thus it is necessary to have the governing equations in a form appropriate for calculating the streamline motion and the corresponding values of h_2 in each region.

GOVERNING DELTA WING STREAMLINE EQUATIONS

Although the directions of the streamlines on the spherical nose are known in advance to be great circles through the stagnation point, additional variables must still be calculated in order to determine h_2 at each location. The equations used directly to do this are available in Reference 5 and are not repeated here.

The following equations for the cylindrical and flat plate areas were derived by Tai[6] as an extension of the work reported in Reference 5. The cylindrical and flat plate regions are shown in Figures 1 and 2.

The governing equations for the cylindrical region are

$$\frac{Dx_b}{DS} = \cos\theta_b \tag{6a}$$

$$\frac{D\phi_b}{DS} = \frac{\sin\theta_b}{R_o} \tag{6b}$$

$$\frac{D\theta_b}{DS} = \frac{1}{\gamma M^2 P}\left(\sin\theta_b \frac{\partial P}{\partial x_b} - \frac{\cos\theta_b}{R_o}\frac{\partial P}{\partial \phi_b}\right) \tag{6c}$$

$$\frac{Dh_2}{DS} = \frac{\partial \theta_b}{\partial \beta}$$

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[6]Tai, T.C., "Streamline Geometry and Equivalent Radius over a Flat Delta Wing with Cyclindrical Leading Edge at Angles of Attack," NSRDC Report 3675 (Oct 1971).

$$\frac{D}{DS}\left(\frac{\partial\theta_b}{\partial\beta}\right) = \frac{1}{\gamma M^2 P}\left(\left(\cos\theta_b \frac{\partial P}{\partial x_b} + \frac{\sin\theta_b}{R_o}\frac{\partial P}{\partial\phi_b}\right)\frac{\partial\theta_b}{\partial\beta}\right.$$

$$- h_2\left(\sin^2\theta_b \frac{\partial^2 P}{\partial x_b{}^2} - \frac{\sin 2\theta_b}{R_o}\frac{\partial^2 P}{\partial\phi_b\partial x_b} + \frac{\cos^2\theta_b}{R_o{}^2}\frac{\partial^2 P}{\partial\phi_b{}^2}\right)$$

$$\left. + \frac{2-M^2}{\gamma M^2 P}\left(\sin\theta_b \frac{\partial P}{\partial x_b} - \frac{\cos\theta_b}{R_o}\frac{\partial P}{\partial\phi_b}\right)^2\right) - h_2\left(\frac{D\theta_b}{DS}\right)^2 \qquad (6e)$$

The line ϕ equals zero corresponds to the zero angle of attack cylindrical stagnation line.

The flat underside of the wing is governed by the following equations

$$\frac{Dx_c}{DS} = \cos\theta_c \qquad (7a)$$

$$\frac{D\phi_c}{DS} = \frac{\sin\theta_c}{x_c} \qquad (7b)$$

$$\frac{D\theta_c}{DS} = \frac{1}{\gamma M^2 P}\left(\sin\theta_c \frac{\partial P}{\partial x_c} - \frac{\cos\theta_c}{x_c}\frac{\partial P}{\partial\phi_c}\right) - \frac{\sin\theta_c}{x_c} \qquad (7c)$$

$$\frac{Dh_2}{DS} = \frac{\partial\theta_c}{\partial\beta} + \frac{h_2\cos\theta_c}{x_c} \qquad (7d)$$

$$\frac{D}{DS}\left(\frac{\partial\theta_c}{\partial\beta}\right) = \left(\frac{1}{\gamma M^2 P}\left(\cos\theta_c \frac{\partial P}{\partial x_c} + \frac{\sin\theta_c}{x_c}\frac{\partial P}{\partial\phi_c}\right) - \frac{\cos\theta_c}{x_c}\right)\frac{\partial\theta_c}{\partial\beta}$$

$$- h_2\left(\frac{D\theta_c}{DS}\left(\frac{D\theta_c}{DS} + \frac{\sin\theta_c}{x_c}\right) + \frac{\sin^2\theta_c}{x_c^2}\right.$$

$$+ \frac{1}{\gamma M^2 P}\left(\sin^2\theta_c \frac{\partial^2 P}{\partial x_c^2} - \sin\theta_c\cos\theta_c\left(\frac{2}{x_c}\frac{\partial^2 P}{\partial\phi_c\partial x_c} - \frac{1}{x_c^2}\frac{\partial P}{\partial\phi_c}\right)\right.$$

$$\left.\left. + \frac{\cos^2\theta_c}{x_c^2}\frac{\partial^2 P}{\partial\phi_c^2}\right) + \frac{2-M^2}{\gamma M^2 P}\left(\sin\theta_c\frac{\partial P}{\partial x_c} - \frac{\cos\theta_c}{x_c}\frac{\partial P}{\partial\phi_c}\right)^2\right) \quad (7e)$$

It is also important to know the joining conditions between the sphere and the cylinder and between the cylinder and the flat plate regions. Figure 3 is a diagram of the sphere region. The streamline that is being analyzed goes off the sphere and onto the cylinder when

$$S/R_o > f \quad (8)$$

where S is the streamline length

R_o is the sphere radius and

f (see Figure 3) is as defined in Equation (10b).

The initial conditions on the cylinder are determined from the final conditions on the sphere and from the coordinates and equations of the cylindrical region.

$$x_{b_i} = 0 \quad (9a)$$

$$\phi_{b_i} = \frac{\pi}{2} - b \quad (9b)$$

$$\theta_{b_i} = E - \frac{\pi}{2} \tag{9c}$$

$$\left(\frac{\partial \theta_b}{\partial \beta}\right)_i = \cos f \tag{9d}$$

E and b are spherical dimensions shown in Figure 3. The unknowns in the preceding development, namely, E and f, may be readily determined from the law of sines and cosines as

$$E = \cos^{-1} \left(-\cos F \cos B + \sin F \sin B \cos e\right) \tag{10a}$$

$$F = \sin^{-1} \left(\sin f \frac{\sin e}{\sin E}\right) \tag{10b}$$

The streamline goes off the cylinder and onto the flat plate region when $\phi_b > 90$ degrees. The initial conditions on the flat plate are

$$x_{c_i} = x_{b_f} \tag{11a}$$

$$\phi_{c_i} = 0 \tag{11b}$$

$$\theta_{c_i} = \theta_{b_f} \tag{11c}$$

$$\frac{\partial \theta}{\partial \beta}_{c_i} = \left(\frac{\partial h_2}{\partial S}\right)_{b_f} - \frac{h_{2_{b_f}} \cos\theta_{b_f}}{x_{b_f}} \tag{11d}$$

DELTA WING PRESSURE DISTRIBUTION EQUATIONS

The pressure distribution must be known on each region of the delta wing in order to apply the governing streamline equations. This section presents various expressions for pressure distribution.

In the spherical nose region, a modified Newtonian pressure distribution is used in the numerical calculations

$$\frac{P}{P_o} = \cos^2\left(\frac{S}{R_o}\right) + \frac{P_\infty}{P_o}\left(1 - \cos^2\left(\frac{S}{R_0}\right)\right) \tag{12}$$

This pressure distribution is a function only of distance from the stagnation point. This formula can be expected to be in greatest error where the sphere merges to the cylinder when the delta wing is at low angles of attack.

For the cylindrical leading edges and flat plate area of the delta wing, the pressure equations are found directly from the theory presented in Buck and McLaughlin.[7] The pressure equation for the cylindrical region is

$$\frac{P}{P_\infty} = C_1\Big(0.320 + 0.455\cos\phi_b' + 0.195\cos 2\phi_b' + 0.035\cos 3\phi_b' - 0.005\cos 4\phi_b'\Big) \tag{13a}$$

$$C_1 = \left[\frac{(\gamma+1)M_\infty^2\cos^2\Lambda_e}{2}\right]^{\frac{\gamma}{\gamma-1}}\left[\frac{\gamma+1}{2\gamma M_\infty^2\cos^2\Lambda_e - (\gamma-1)}\right]^{\frac{1}{\gamma-1}} \tag{13b}$$

where Λ_e is the effective sweep angle of the wing as defined in Reference 7, and ϕ_b' is a cylindrical angle measured with respect to the effective leading-edge stagnation line.

Note that the previous expression for pressure distribution is a coordinate function solely of ϕ_b'. Equations (13) are based on correlated cylinder data. The cylindrical pressure equations include corrections for the angle of attack and sweep, the neglect of the nose region, and the neglect of the afterbody effect of the flat plate area, that is, the cylinder does not continue through after the point where $\phi_b = 90$ degrees. This

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[7] Buck, M.L. and McLaughlin, E.J., "A Technique for Predicting Pressure Distributions and Aerodynamic Force Coefficients for Hypersonic Winged Re-Entry Vehicles (U)," Wright-Patterson Air Force Base, Aeronautical Systems Division, TDR-63-522 (Aug 1963). CONFIDENTIAL

last mentioned factor is considered to account for the small pressure jump between the preceding cylindrical formula and the flat plate formula shown as follows.

The expression for pressure on the windward flat plate region is

$$\frac{P}{P_\infty} = \left(1 + \frac{\gamma}{2} KM_\infty{}^2 \sin^2\alpha\right) + BC_\gamma \left(\frac{C_D}{x_n+\Delta}\right)^{2/3} M_\infty{}^2\cos^2\Lambda \tag{14}$$

The first right-hand-side term is a Newtonian-type expression. K is a function of angle of attack; K versus α is given in Reference 7 where theoretical results are compared with flat plate experimental data. The second small additional term is what creates the pressure variation on the flat plate region. This term is based on an analogous blast-wave-type effect across the blunt cylindrical leading edge. The most important fact to note in the second expression is that the pressure change produced is normal to the leading edge, x_n-direction. The constants appearing in this expression are given in Reference 7. The present analysis does not consider boundary layer-displacement effects on the pressure.

It is difficult to determine the value of Δ, the shock standoff distance along the leading edge. In the computer program, it was set equal to the standoff distance in front of the spherical nose. Pressure values determined by other theories may be readily substituted into the program in place of the preceding results, provided that the derivatives of pressure can be explicitly evaluated. Finally, it should be noted that in the vicinity of the centerline when

$$\frac{\pi/2 - \Lambda - \phi_c}{\pi/2 - \Lambda} < 0.06 \tag{15}$$

the pressure derivatives have been set equal to zero in the program.

THERMODYNAMIC VARIABLES

The upstream flow values M_∞, T_o, P_o, μ_o, Pr, C_p, and γ are known. The boundary layer is assumed to be formed of upstream flow that has passed through a normal shock wave in front of the spherical nose cap; this point

will be discussed further in a later section. The stagnation pressure P_o of the flow entering the boundary layer is therefore

$$P_o = P_{oo} \left[\frac{(\gamma+1)M_\infty^2}{(\gamma+1)M_\infty^2 - 2} \right]^{\frac{\gamma}{\gamma-1}} \left[\frac{\gamma+1}{2\gamma M_\infty^2 - (\gamma-1)} \right]^{\frac{1}{\gamma-1}} \tag{16}$$

The value of pressure in the boundary layer is taken as equal to the inviscid pressure at the edge of the boundary layer. The inviscid pressure at each computed point is determined from Newtonian or other theories that will be described. The local Mach number on the surface of the body (for flow that has passed through the shock) is then found from the isentropic flow equation

$$M = \left\{ \frac{2}{\gamma-1} \left[\left(\frac{P}{P_o} \right)^{-\frac{\gamma-1}{\gamma}} - 1 \right] \right\}^{1/2} \tag{17}$$

To determine the heat transfer, various thermodynamic values must first be determined at infinity, at the edge of the boundary layer, at reference conditions, and at the wall. The four heat transfer equations used in the analysis can be written in terms of these values.

The flow values in each region are readily calculated. At infinity upstream, the necessary flow values are

$$P_\infty = \frac{P_{oo}}{\left(1 + \frac{\gamma-1}{2} M_\infty^2\right)^{\frac{\gamma}{\gamma-1}}} \tag{18a}$$

$$T_\infty = \frac{T_o}{\left(1 + \frac{\gamma-1}{2} M_\infty^2\right)} \tag{18b}$$

$$\rho_\infty = \frac{P_\infty}{R_G T_\infty} \tag{18c}$$

$$u_\infty = M_\infty \left(32.2\ \gamma\ R_G T_\infty\right)^{0.5} (3600) \tag{18d}$$

$$\mu_\infty = \mu_o \left(\frac{T_\infty}{T_o}\right)^{1.5} \left(\frac{T_e + 110}{T_\infty + 110}\right) \tag{18e}$$

The preceding inviscid equations are from Shapiro[8] and the National Advisory Committee for Aeronautics-Ames tables.[9] The Sutherland law, as given in Schlichting,[10] was used for the variation of viscosity with temperature. At the boundary layer edge, the flow values become

$$\rho_e = \frac{P_e}{R_G T_e} \tag{19a}$$

$$T_e = \left(\frac{P_e}{P_o}\right)^{\frac{\gamma-1}{\gamma}} \left(T_o\right) \tag{19b}$$

$$\mu_e = \mu_o \left(\frac{T_e}{T_o}\right)^{1.5} \left(\frac{T_o + 110}{T_e + 110}\right) \tag{19c}$$

$$u_e = M \left(32.2\ \gamma\ R_G T_e\right)^{0.5}\ 3600 \tag{19d}$$

English engineering units are used with time in hours, and therefore the calculations of velocity values are performed in the program with a conversion factor of 3600. Finally, the reference conditions are

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[8] Shapiro, A.H., "The Dynamics and Thermodynamics of Compressible Fluid Flow." Vol. 1. Ronald Press, New York, (1953).

[9] National Advisory Committee for Aeronautics, "Equations, Tables, and Charts for Compressible Flow," NACA Report 1135 (1953).

[10] Schlichting, H., "Boundary-Layer Theory," Sixth Edition McGraw-Hill, Inc., New York (1968).

$$T_r = 0.5T_e + 0.5T_w + 0.22\left(T_{aw} - T_e\right) \tag{20a}$$

$$\mu_r = \mu_o \left(\frac{T_r}{T_o}\right)^{1.5} \left(\frac{T_o + 110}{T_e + 110}\right) \tag{20b}$$

$$\rho_r = \frac{P_e}{R_G T_r} \tag{20c}$$

$$\nu_r = \frac{\mu_r}{\rho_r} \tag{20d}$$

where

$$T_{aw} = T_e + \frac{\left(r_c\, M^2\, \gamma\, R_G T_e\right)}{2Cp \cdot 778} \tag{20e}$$

and r_c is the recovery factor set equal to $P_r^{1/2}$ in laminar flow and to $P_r^{1/3}$ in turbulent flow.

The wall temperature T_w is read into the program at a constant value. The Eckert[11] value of reference temperature is used in the majority of the calculations for obtaining the reference values. It is felt that best overall results will be obtained by using a compromise temperature value as previously described. Vaglio-Laurin, however, suggests other values for reference temperature, and some calculations are subsequently discussed using this approach.

Enthalpy terms required in the heat transfer expressions are: stagnation enthalpy ($H_o = C_p \cdot T_o$), wall enthalpy ($H_w = C_p \cdot T_w$), and adiabatic wall enthalpy ($H_{aw} = C_p \cdot T_{aw}$). Heat transfer values are found in the formulas to be presented using both the terms ($H_o - H_w$) and ($H_{aw} - H_w$). The

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[11] Rohsenow, W.M. and H.Y. Choi., "Heat, Mass, and Momentum Transfer," Prentice Hall, Englewood Cliffs, N.J. (1961).

expression $(H_{aw} - H_w)$ is based on local enthalpy values and is felt to give the best heat transfer results. Finally, it should be noted that γ is common between all subroutines and that its value may be changed in the course of a computation to represent real gas effects. (This was not done in any of the present calculations).

HEAT TRANSFER EQUATIONS

Utilizing the previously found thermodynamic values, the equation for heat transfer using the Lees method[3] for laminar flow in the vicinity of the stagnation point is:

$$q_w = \frac{q_{wo} \left(\frac{P_e}{P_o}\right) \frac{u_e}{u_\infty} h_2 R_o^{1/2}}{\left(\int_o^s \frac{P_e}{P_o} \frac{u_e}{u_\infty} h_2^2 \, dS\right)^{1/2} \cdot 2G} \tag{21}$$

where

$$G = \left[\frac{\gamma-1}{\gamma}\left(1 + \frac{2}{(\gamma-1)M_\infty^2}\right)\left(1 - \frac{1}{\gamma M_\infty^2}\right)\right]^{1/4} \tag{22}$$

The preceding heat transfer equations are derived by analogy with usual axisymmetric formulas; in the axisymmetric formulas, the local radius r is replaced by h_2. The integrals appearing in this and other heat transfer methods are calculated in the program by a Simpson rule subroutine. The integral appearing in the denominator of the Lees method is evaluated in two ways: (1) by forcing it to have a value such that $q_w = q_{wo}$ at the stagnation point, and (2) by initially setting it equal to zero. After the first few integration steps (where the integration method is still unstable in any case), both approaches produce approximately equivalent results.

In the Lees method, q_{wo} is the stagnation point heat transfer. The value of this quantity is calculated from a Stanton number read into the program. The Lees method is a stagnation region solution that results in a decay in heat transfer from the stagnation value previously described. The method therefore has greatest validity in the region in the vicinity of

stagnation point while the flow is still laminar. Other techniques are more appropriate for points further downstream.

A second technique for laminar flow is given in the Vaglio-Laurin paper on laminar streamline divergence.[1]

$$q_w = 0.333 \left(\rho_r \mu_r\right)^{1/2} \left(H-H_w\right) P_r^{-2/3} \left(\frac{u_e}{X_L}\right)^{1/2} \qquad (23a)$$

$$X_L = \frac{1}{\rho_r \mu_r u_e h_2^{\,2}} \int_o^S \rho_r \mu_r u_e h_2^{\,2} \, dS \qquad (23b)$$

The preceding expressions are again based on the h_2 - r analogy. The calculation in the program, using the Vaglio-Laurin laminar method, is performed using two values for enthalpy, $H = H_{aw}$ and $H = H_o$. The Vaglio-Laurin laminar flow method is to be used in laminar flow beginning approximately 20 degrees from the stagnation point.

Two techniques are included in the program for the case of turbulent flow. The first is a streamline divergence technique[4] where

$$q_w = 0.0296 \, P_r^{-2/3} \left(H-H_w\right)\left(u_e \rho_r \mu_r\right)^{1/4} \left(\frac{1}{X_T^{\,1/5}}\right) \qquad (24a)$$

$$X_T = \frac{1}{\rho_r \mu_r u_e h_2^{\,5/4}} \int_o^S \rho_r \mu_r u_e h_2^{\,5/4} \, dS \qquad (24b)$$

It is also derived by analogy with the usual heat transfer method on an axisymmetric body and evaluated here using $H = H_o$ and $H = H_{aw}$. The second technique is the method of Rose et al. as modified by Tai[5] to include pressure gradients effects. This formula is

$$q_w = \frac{0.0296\ P_r^{-2/3}\ (H_o-H_w)\ \rho_e^{1.05}\ \mu_e^{0.8}\ \nu_r^{0.05}\ h_2^{1/4}}{\mu_o^{0.6}\left(\int_o^s \rho_e^{5/4}\ u_e\ \nu_r^{1/4}\ h_2^{5/4}\ dS\right)^{1/5}} \tag{25}$$

In the program, results are determined as the ratio of the local Stanton number to the Stanton number at the stagnation point.

The preceding equations along with expressions for pressure distribution presented previously are used in the computer program to determine heat transfer results along calculated streamline paths. The computer program is not included in this report (but a listing will be kept on file). The program has been annotated so as to describe the operations in each section. A separate sheet is maintained to describe input requirements and control statements. Subroutine form has been used to perform the calculations in the sphere, cylinder, and flat plate regions; thermodynamic values are evaluated in groups (boundary layer edge values, reference values, etc.) and the heat transfer equations are written in terms of this common set of expressions. An attempt has been made to make the program easy to follow; comment cards have been included where appropriate, and variable names have been made to correspond to the spelling of physical terms.

RESULTS AND DISCUSSION

Figures 4-11 compare the results of the calculations, using the previously described theory with experimental delta wing data reported in the literature. The pressure distributions developed in the previous section are first compared with the measured pressure values. Then after a discussion of resulting streamline patterns, comparsion is made between computed and measured heat transfer values.

For laminar flow, results obtained by using the computer program are compared with experimental National Aeronautics and Space Administration data on a 70-degree slab, delta wing (with the same basic geometry as used in the present analysis) in hypersonic flow at several Mach numbers and various angles of attack.

As seen in Figure 4, agreement was satisfactorily obtained for the spherical nose region, pressure distribution. For the most part, agreement of computed and measured[12] pressure values was good at various positions normal to the leading edge; see Figure 5. At high angles of attack, the flat plate part of the delta wing will strongly influence the pressure distribution on the cylindrical leading edge, and this tends to make the correlated cylindrical leading-edge pressure formula in Equation (13a) invalid. The comparsion of experimental and theoretical pressures along the centerline of the wing at various angles of attack (Figure 6) is satisfactory.

Figure 7 shows computer-generated streamline patterns on the same wing, and Figure 8 gives the corresponding experimental results. At low angles of attack, the computed streamline patterns on the flat part of the delta wing are in qualitative agreement with the experimental data. (Comparsion is possible only on the flat part of the wing.) At higher angles of attack, the experimental streamlines tend to go outward. A possible explanation of this effect is that pressure may increase on the flat plate region in the ϕ_c-direction (pressure would be highest on the centerline). This would tend to make the streamlines go outward. On this assumption, an option has now been put into the program so that pressure can be increased proportionally with ϕ_c on the flat plate area. In certain cases, however, the streamlines will also develop a tendency to cross; this is attributed to the use of an inadequate pressure gradient.

Analysis of NASA pressure data indicates that at least in this case study, the pressure did not increase when going toward the centerline. It is then realized that the streamline pattern on the flat plate underside of the wing depends on two factors: (1) the pressure distribution and (2) the direction with which the streamline merges onto the flat plate region. The latter is particularly affected by pressure gradient on the cylindrical region.

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[12] Bertram, M.H. and P.E. Everhart, "An Experimental Study of the Pressure and Heat-Transfer Distribution on a 70° Sweep Slab Delta Wing in Hypersonic Flow," National Aeronautics and Space Administration Technical Report R-153 (1963).

A Newtonian conical flow, flat-plate pressure distribution has been incorporated into the program as an alternative means of computing the pressure distribution in this area. The theory here is taken from References 13 and 14, although it should be noted that this theoretical development assumes sharp leading-edge wings. The pressure and streamline patterns computed by these formulas are not radically different from those previously discussed.

The correctness of the heat transfer-calculation methods (in both the laminar cases now being discussed and the turbulent cases given later) is determined by studying the spherical nose, since one can be most certain about the pressure distribution in that area. The heat transfer results on the spherical nose are shown in Figure 9. After the first few computed time intervals, both calculation procedures using the Lees method give equivalent results, and these have therefore been shown as a single line. Results are also shown for the Vaglio-Laurin laminar-streamline divergence method. The Vaglio-Laurin results using $(H_{aw}-H_w)$ fall lower than the Lees results at the higher angles of attack. This trend is believed to be correct, and, in general, the agreement is considered good.

Figure 10 shows computed and experimental heat transfer results for the cylindrical leading edge and flat plate regions at angles of attack of α = 20 and 30 degrees. Data are plotted with respect to a line normal to the leading line, following the surface contour. The computed values are reasonable and correct in trend but slightly higher (more so for the 20-degree case) than the experimental values. Additional discussion of these results will be presented subsequently. Calculations have been made at higher values of angle of attack, 40 and 45 degrees; the basic trend of these results (decreasing Stanton number along a streamline) for regions downstream on the flat plate segments of the wing disagreed with experimental values which showed an approximately constant Stanton number. The theories

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[13] Hida, K., "Thickness Effect on the Force of Slender Delta Wings in Hypersonic Flow," American Institute of Aeronautic and Astronautics Journal New York, pp. 427-433 (Mar 1965).

[14] Polak, A. and T. Y. Li, "Three-Dimensional Boundary-Layer Flow over a Flat Delta Wing at a Moderate Angle of Attack," American Institute of Aeronautics and Astronautics Journal New York, pp. 233-240 (Feb 1967).

presented in Reference 12 for the wing centerline predict approximately the same incorrect results downstream. This discrepancy is believed to be due to differences in the basic streamline pattern at these higher angles of attack as already discussed. At lower angles of attack, agreement between computed results and theory is satisfactory on the cylindrical region; however, there is some disagreement for the flat plate part. This difference could be partially explained by limitations in the theory itself. As will be discussed further, the theory is limited to the nose regions since the external flow has not yet expanded to a high velocity in that area. (The theory breaks when $u_e^2/(H_e-H_w)$ is not much less than 1.) It can then be noted that the greatest external flow velocities u_e occur on the aft part of the flat plate region at low angles of attack.

Results presented in Figure 10 are essentially heat transfer values plotted along lines normal to the leading edge. (This represents a reevaluation of the printout data which are in the form of heat transfer values along the streamlines.) Plotted in this manner, the computed heat transfer results indicate a falloff in heat transfer values when moving outward along the span, constant s/d, on the cylindrical leading edge; in contrast, the heat transfer values seem to fall approximately on a single curve on the flat plate region. This type of information could be useful for design purposes.

Finally, for the laminar flow case, calculations were also performed at α = 20 degrees, using the laminar reference values $\rho_r\mu_r=(\rho_e\mu_e)^{0.8}(\rho_w\mu_w)^{0.2}$ (an option in the program) suggested by Vaglio-Laurin. However, these results were not as good as those obtained using the Eckert reference values.

For the case turbulent flow, the correctness of the heat transfer methods was determined by comparsion (matching data at the maximum point) with hemispherical nose data of Thomas et al.[15] All the various techniques used seemed to give reasonable results.

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[15] Thomas, A. C. et al., "Advance Re-Entry Systems Heat Transfer Manual for Hypersonic Flight," Wright-Patterson Air Force Base, Flight Dynamics Laboratory Technical Report 65-195, Boeing Co. D2-85029-1. Contract AF33(657)7132 (Oct 1966).

The computed heat transfer results for the case of turbulent flow were also compared with NASA experimental values for the type of delta wing under consideration.[16] Only a few sets of data giving both pressure distribution and heat transfer are presented, and, unfortunately, the reported heat transfer results are limited. Comparsion was made with a delta wing swept back 73 degrees at $M_\infty = 7$, $\alpha = 20$ and 30 degrees (30 degrees was the highest value reported). The results (Figure 11) tended to be somewhat similar to the laminar case in that agreement was best for the 30-degree example. As was found by Tai,[5] results are better for the modified technique of Rose et al. than for the streamwise divergence, turbulent flow method. Only centerline turbulent flow data are reported in Reference 16, whereas the present program calculates streamlines going about the body (from the spherical nose to the cylindrical leading edge to the flat plate area) which eventually intersect the centerline. Because of this difference in calculation procedures, the present comparsion is actually a worst-case type of comparsion.

In the development of his theory, Vaglio-Laurin states various requirements:

1. The flow must be hypersonic
2. The wall must be cold and
3. $u_e^2/(H_e-H_w)$ must be very much less than 1.

Basically, these requirements are needed to ensure that the boundary layer flow direction will be the same as the external flow direction; this would not be the case, for instance, in the subsonic or transonic regimes. Requirement 3 indicates that the validity of the theory will decrease as one moves downstream of the nose onto regions of the body that exhibit a low effective angle of attack relative to the free stream. In view of this, the validity of the theory at a given point on the body may be expected to increase with increasing angle of attack. The pressure formulas used in the present study, however, lose validity at the higher angle of attack values. It is therefore not unexpected that the agreement between theory and data maximizes at a moderate value of angle of attack, 30 degrees.

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[16]Nagel, A. L. et al., "Analysis of Hypersonic Pressure and Heat Transfer Tests on Delta Wings with Laminar and Turbulent Boundary Layers," National Aeronautics and Space Administration Contractor Report 535 Boeing Co. Contract NAS 1-4301, (Aug 1966).

A second point to be discussed is the influence of the shock wave shape characteristics for the delta wing. The important point here is that the boundary layer-edge flow (which is eventually entrained into the boundary layer) is calculated as an isentropic expansion of the flow downstream of the shock wave. The present program assumes that all the flow that is entrained into the boundary layer has passed through the normal shock at the nose; this, of course, is an approximation, especially in the case of thin wings at low angles of attack. A second approach is to initiate the expansion across an oblique shock determined by the flat plate part of the wing. The flow is between the two extremes.

The feature that characterizes three-dimensional boundary layers is the fact that the streamline path is to be evaluated as part of the analysis. This is in contrast to the axisymmetric case where the streamline path is determined completely by symmetry. For this reason, the influence of pressure distribution on the heat transfer characteristics is doubly cumulative in three-dimensional flows--pressure influences both streamline path and local thermodynamic variables. Better pressure distributions should produce more accurate heat transfer values.

CONCLUSION

A computer program has been generated to calculate the heat transfer characteristics on the windward side of a delta wing in hypersonic flow. The program utilizes the small crossflow theory of Vaglio-Laurin which is limited to hypersonic speeds, cold walls, and moderate boundary layer-edge velocities. This program may be used either for analyzing the thermal environment on a proposed configuration or for studying the parameters that can effect heat transfer for a three-dimensional configuration.

Comparsions of computed results and experimental delta wing heat transfer data appearing in the literature show reasonable agreement, especially for cases where the limitations of the theory are not exceeded, and the pressure distributions used as input are most accurate.

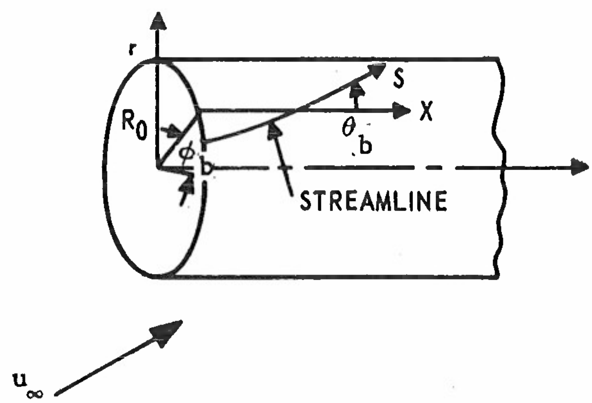


Figure 1 - Cylindrical Leading Edge Geometry

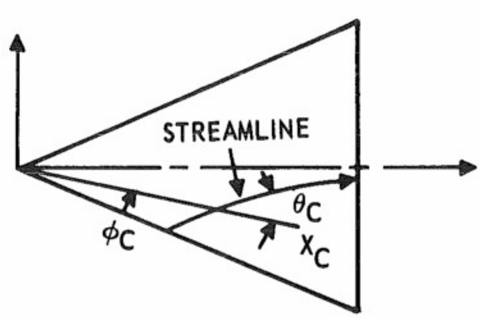


Figure 2 - Lower Wing Surface

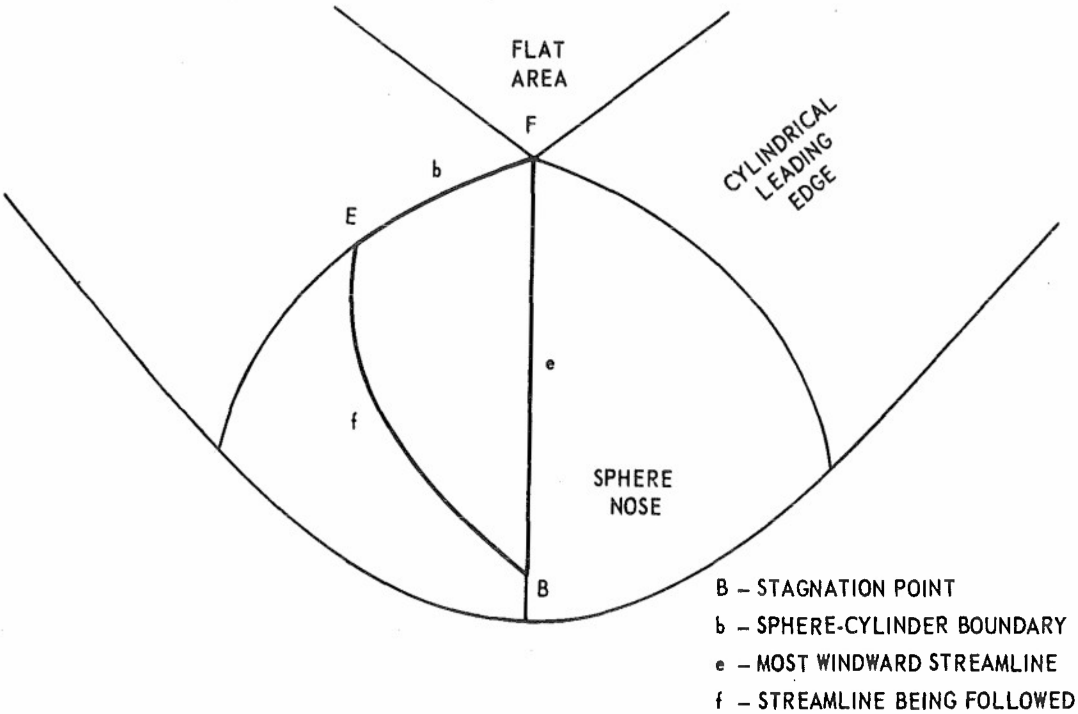


Figure 3 - Spherical Nose Part of Delta Wing

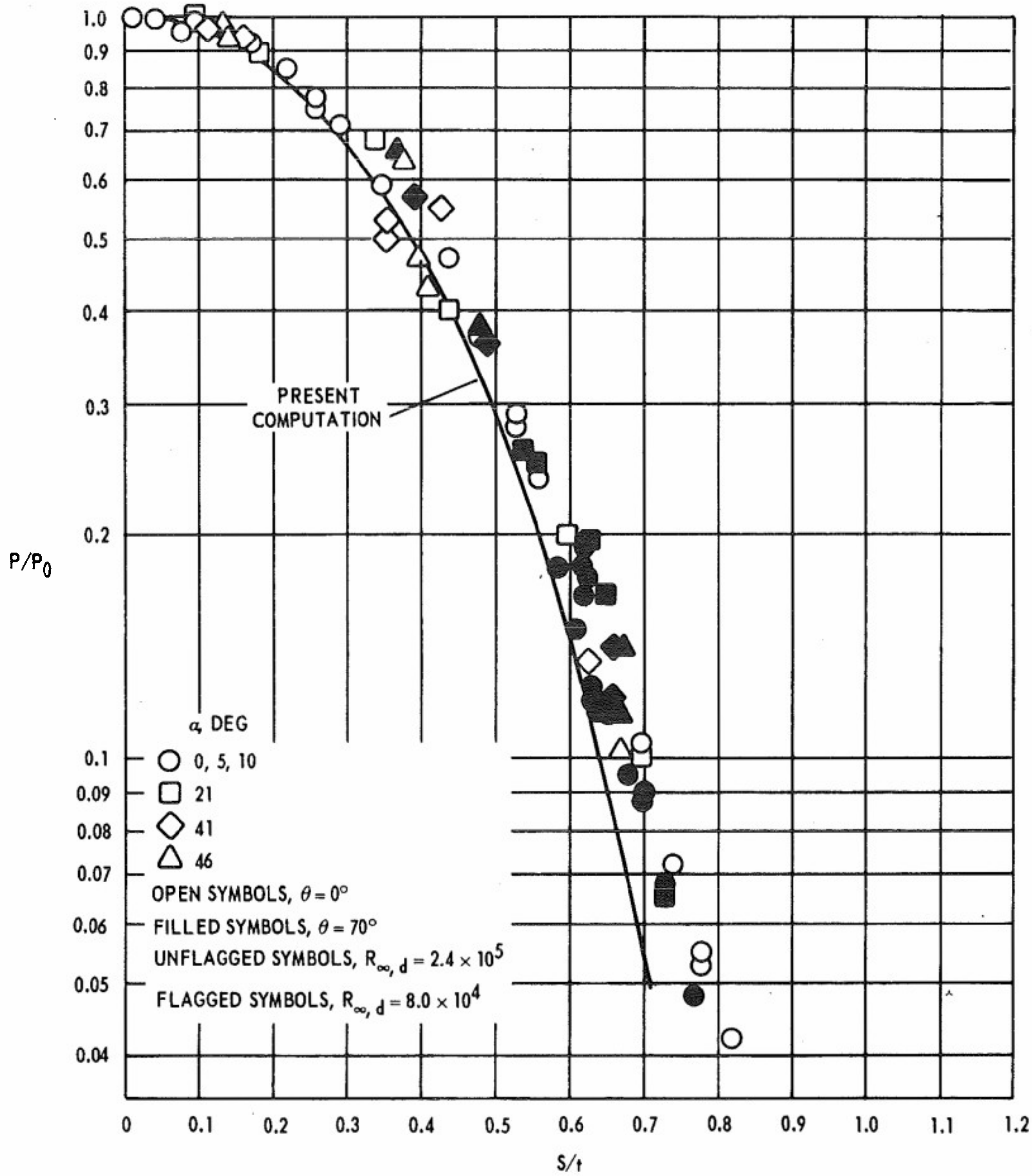


Figure 4 - P/P_o on Spherical Nose
(Measured values are from Reference 12)

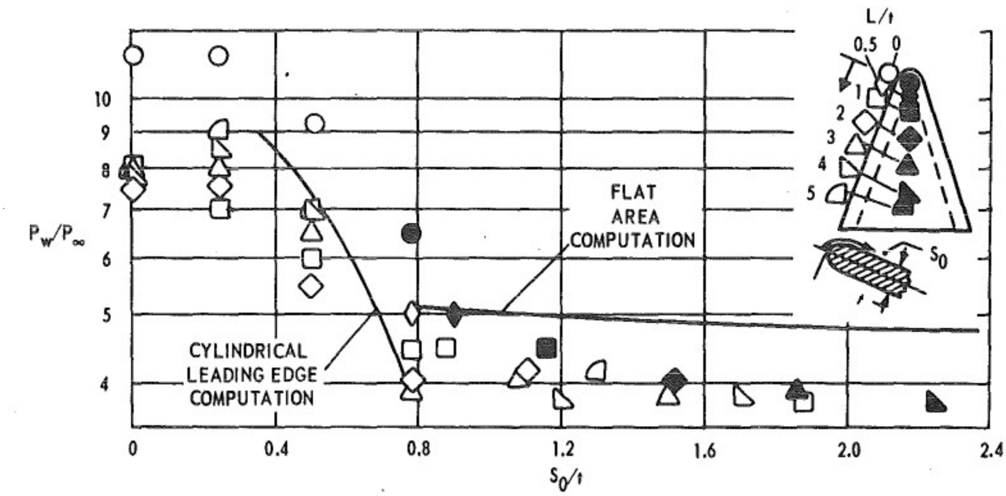


Figure 5a - α ≅ 10 Degrees

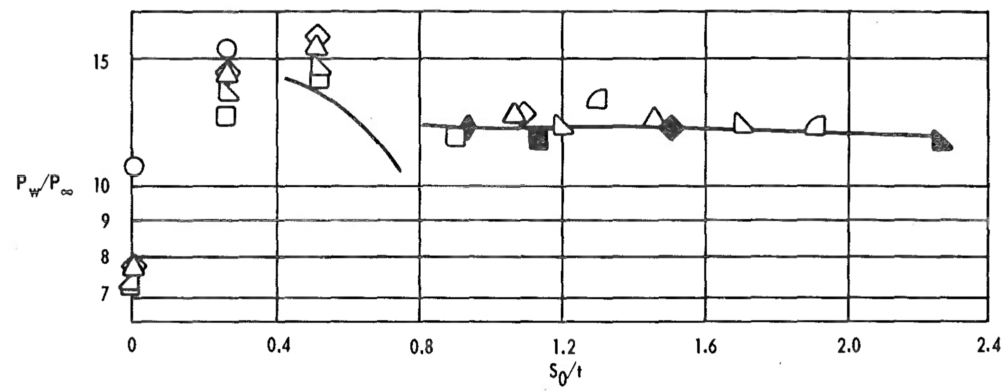


Figure 5b - α ≅ 21 Degrees

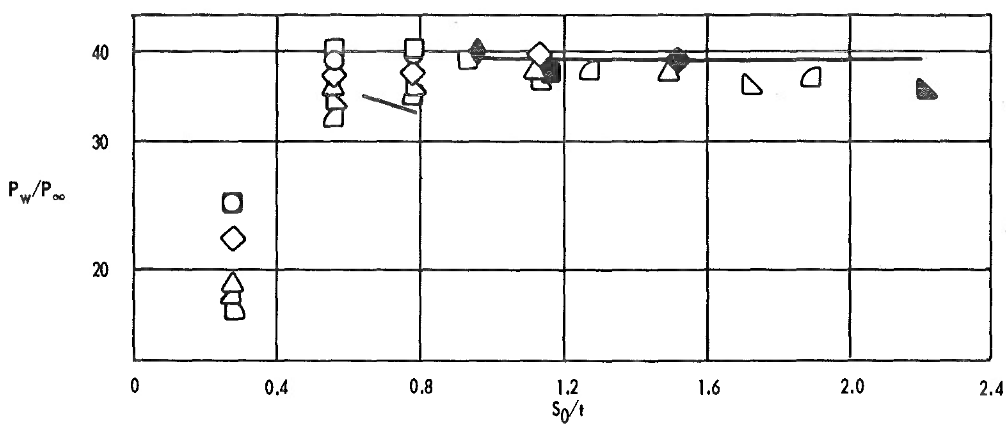


Figure 5c - α ≅ 46 Degrees

Figure 5 - Pressure Distribution along Lines Normal to Wing Cylindrical Leading Edge M_∞ = 6.8

(Lines shown are present computation)

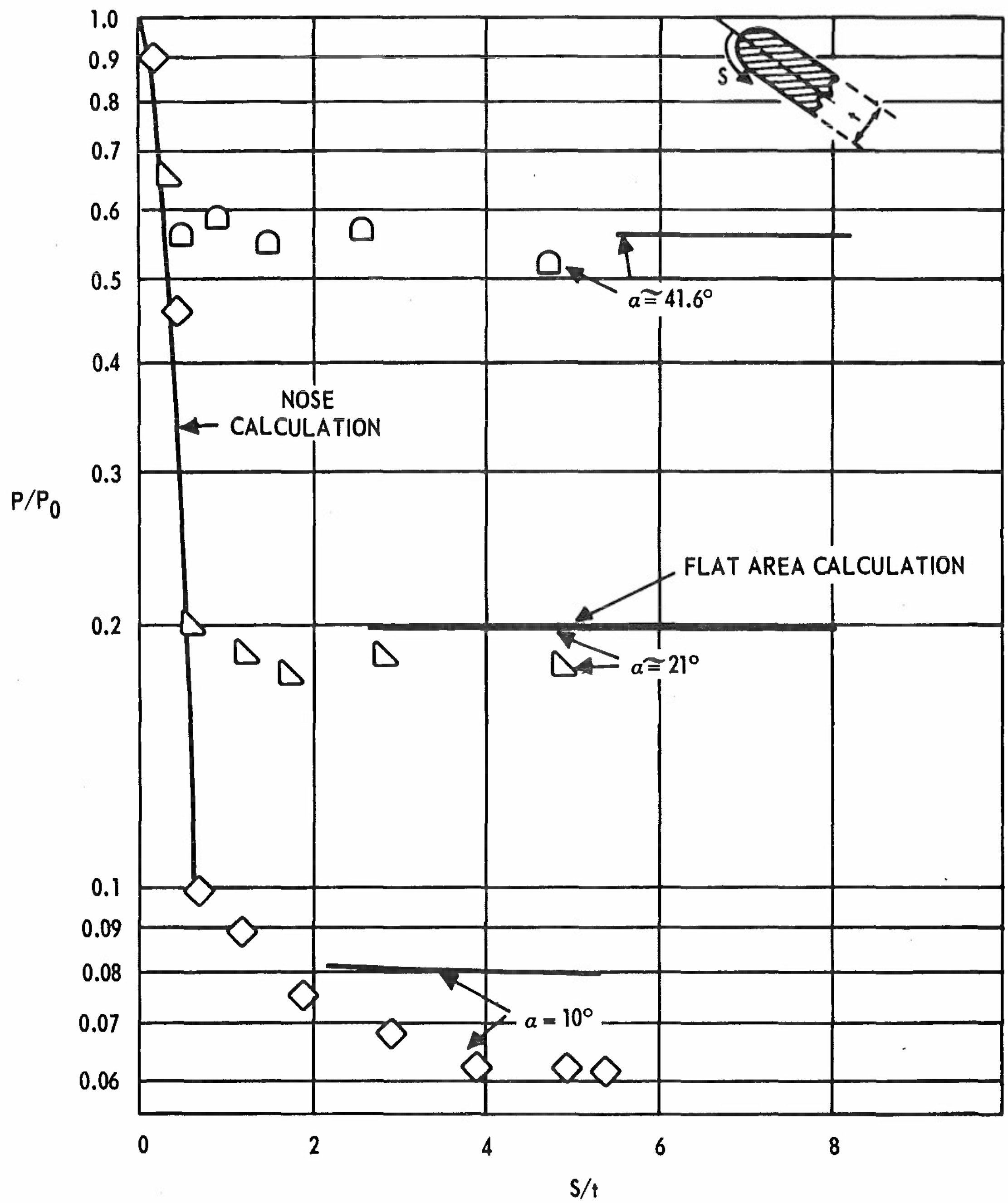
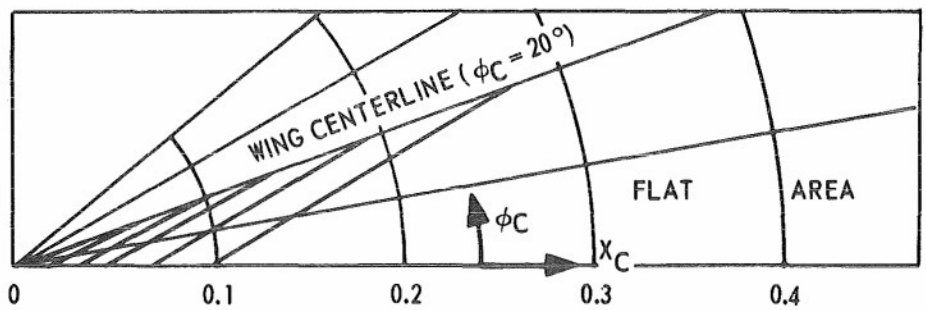


Figure 6 - Centerline Pressure Distribution
(Experimental values are from Reference 12)

Figure 7 - Computed Streamline Patterns on Cylindrical
and Flat Plate Regions



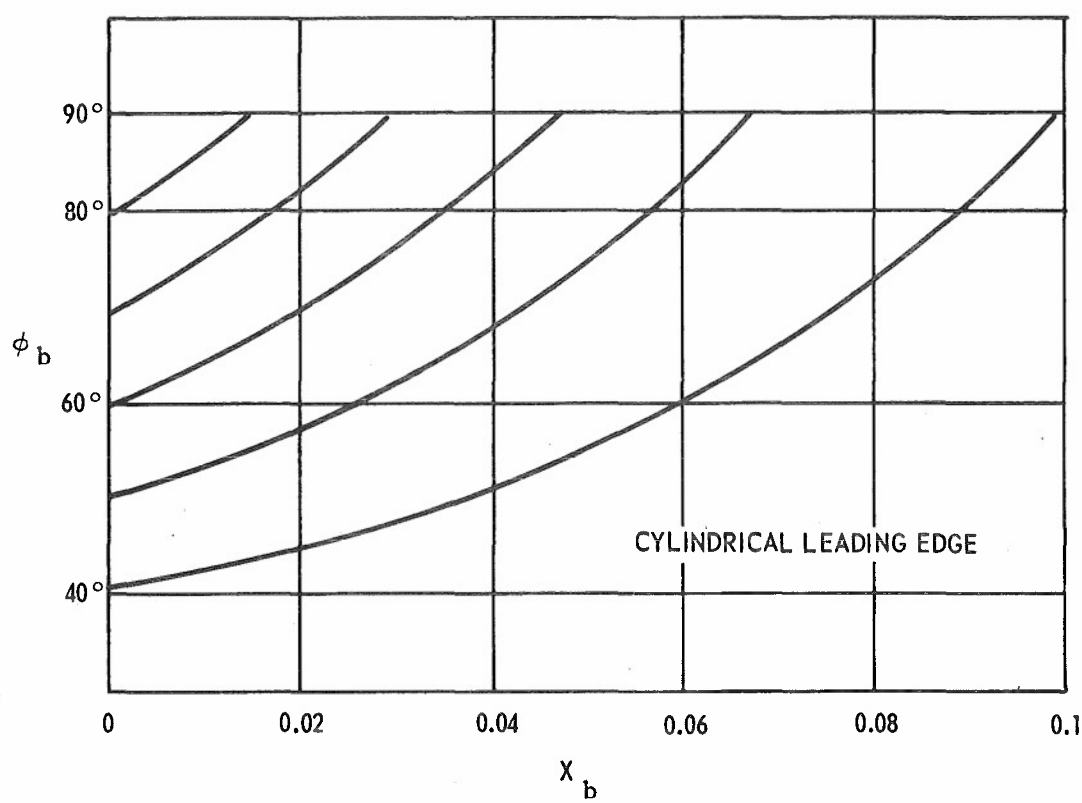
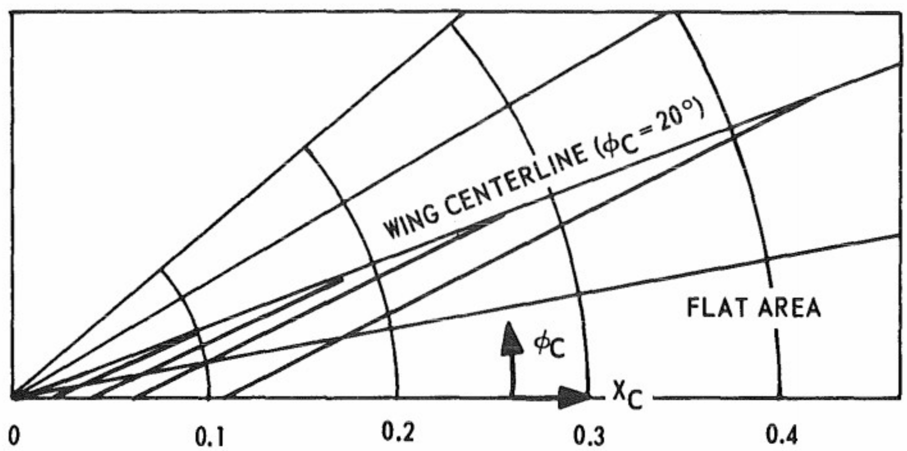


Figure 7a - α = 10 Degrees



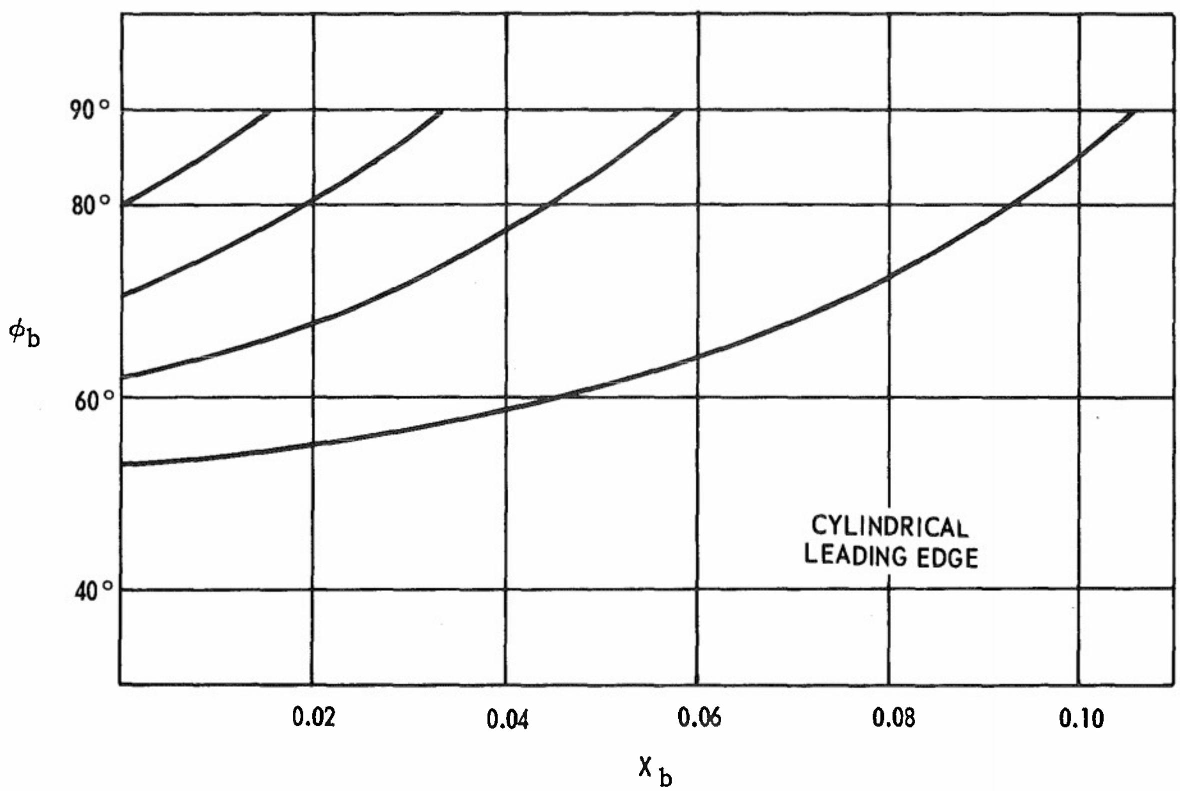
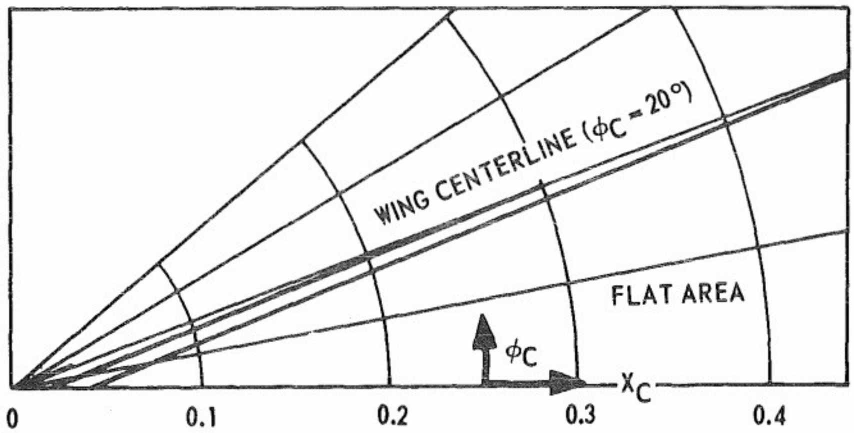


Figure 7b - $\alpha \cong 21$ Degrees



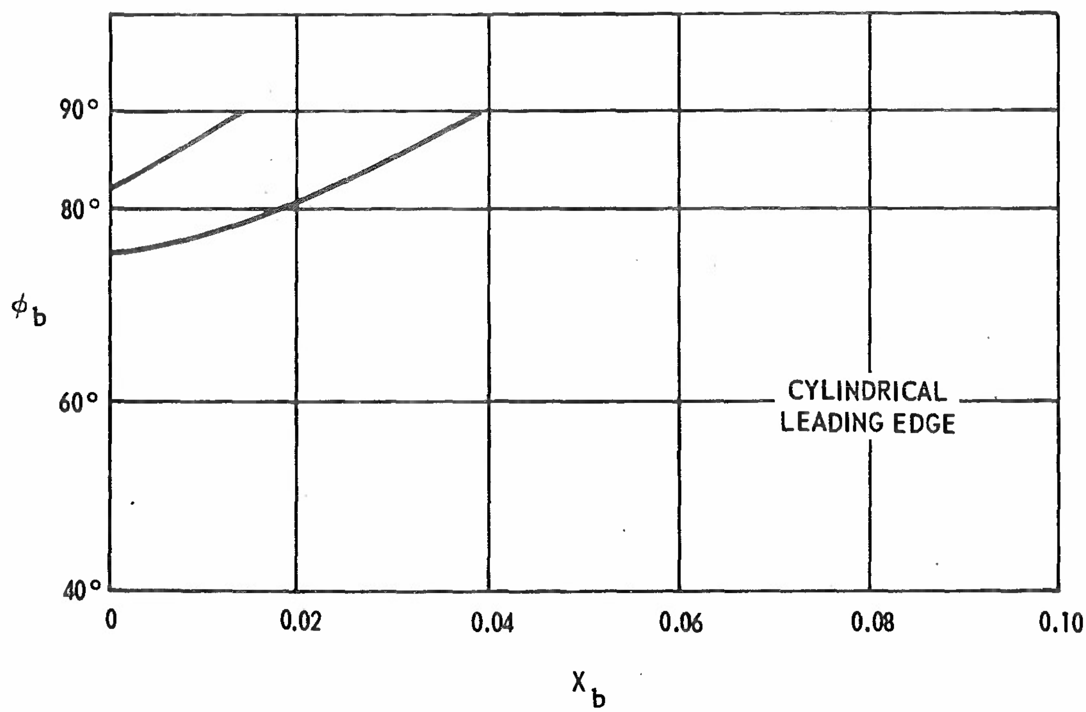


Figure 7c - α = 41.6 Degrees

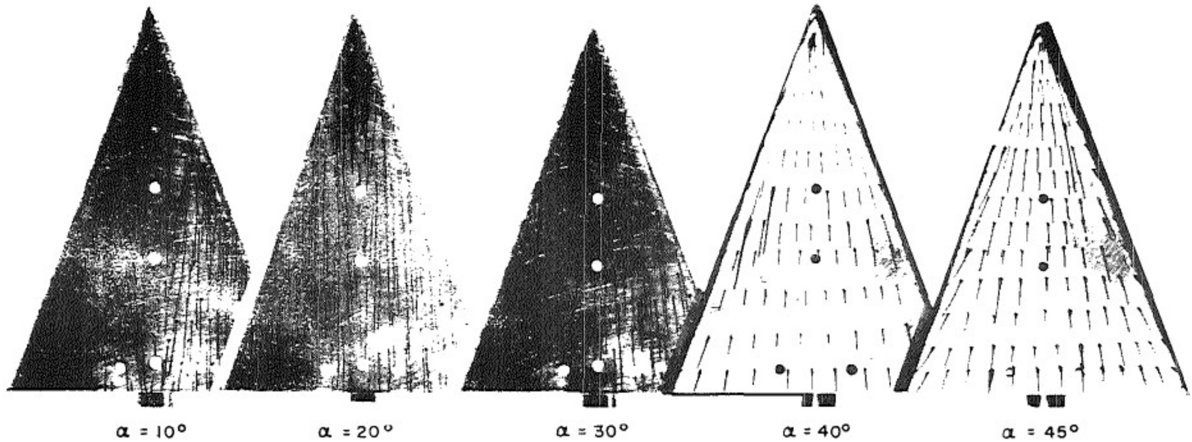


Figure 8 - Surface Flow Streamline Patterns at Several Angles of Attack
(From Reference 12; $M_\infty = 6.9$)

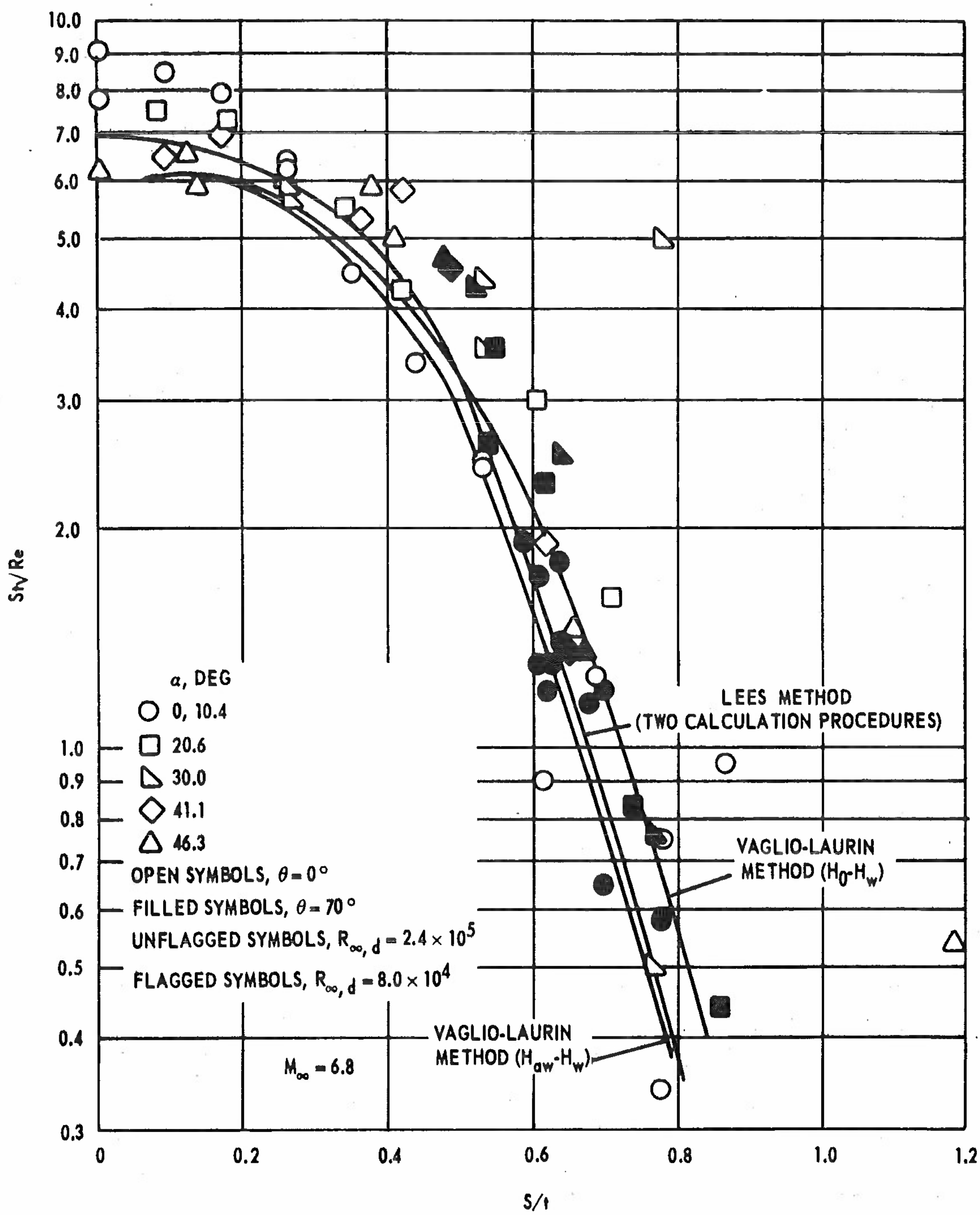


Figure 9 - Heat Transfer Distribution on Spherical Nose
(Laminar Flow)

(Experimental values are from Reference 12)

Figure 10 - Stanton Number Distribution along Lines Normal to Wing Cylindrical Leading Edge $M_\infty = 6.8$

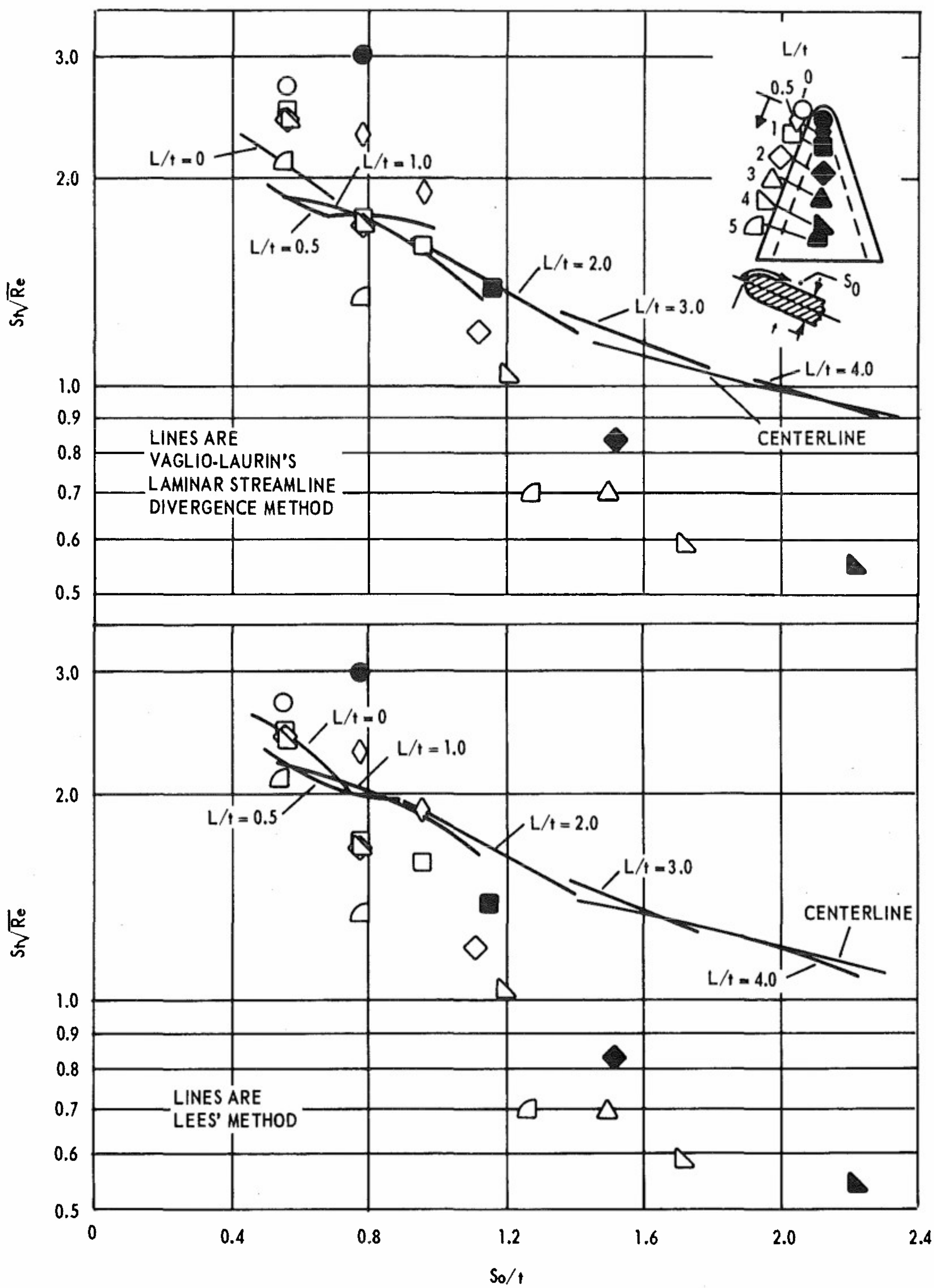


Figure 10a - α = 20.6 Degrees

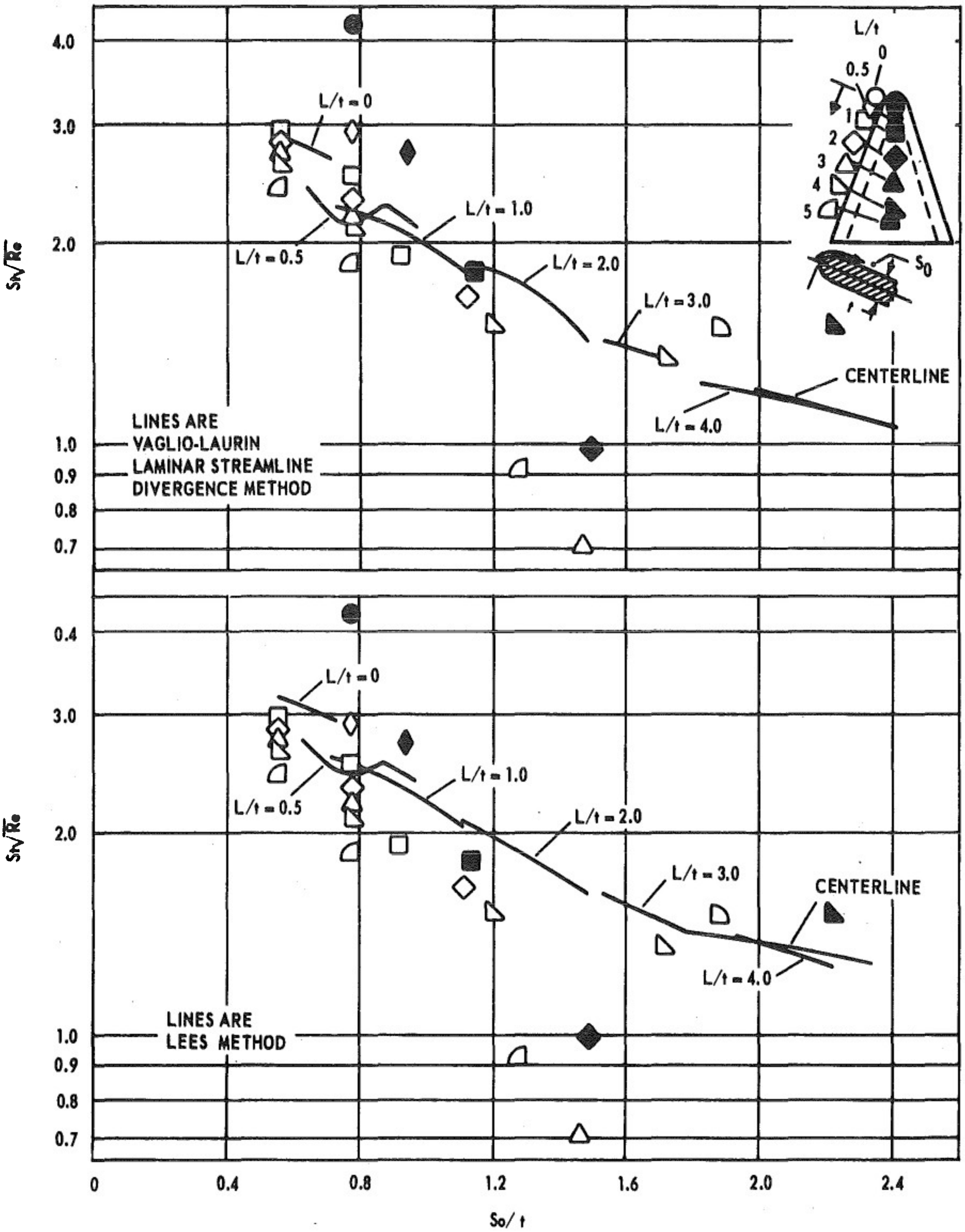


Figure 10b - α = 30 Degrees

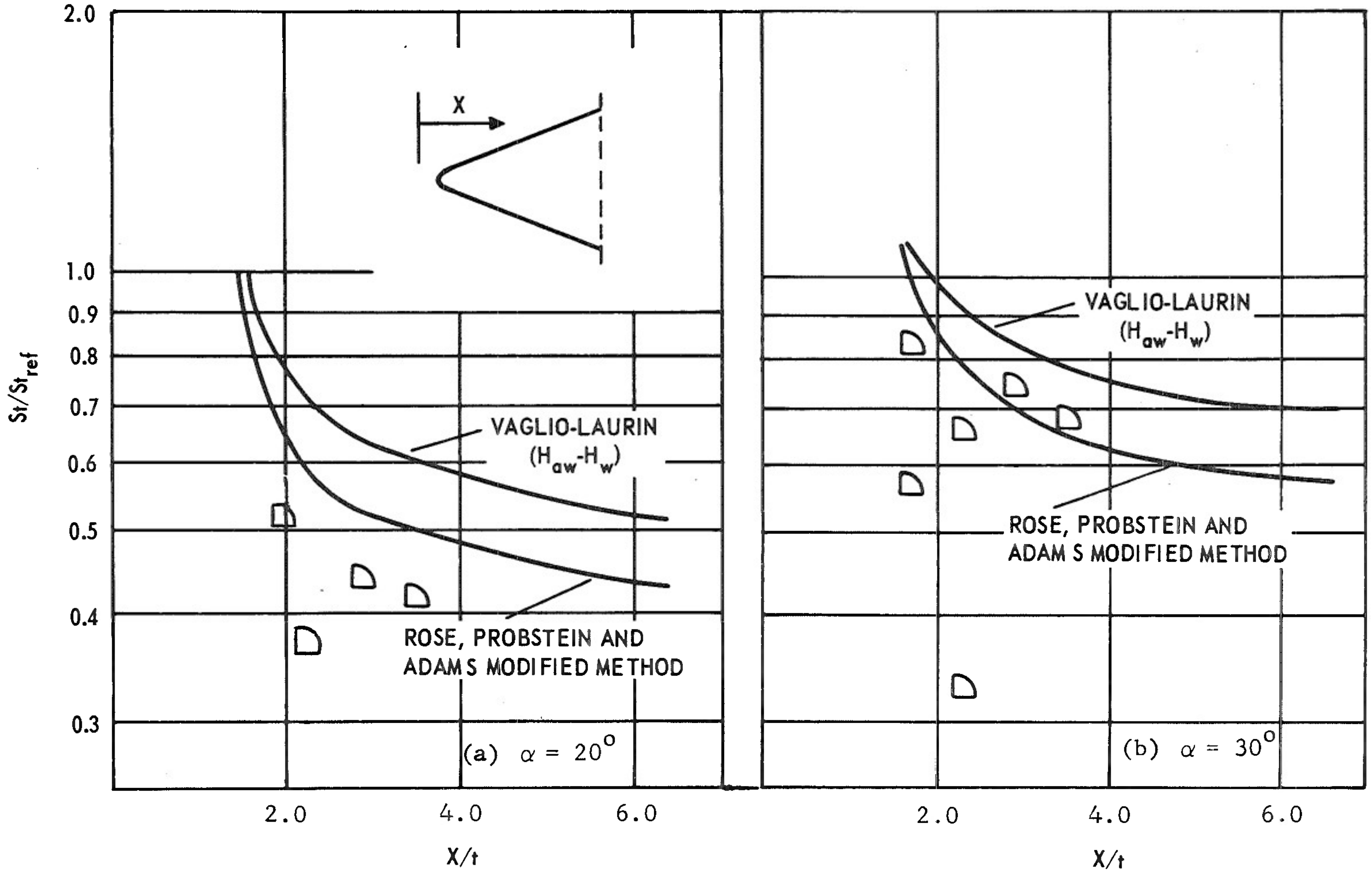


Figure 11 - Heat Transfer Distribution on Centerline for Delta Wing with Turbulent Boundary Layer

(Data from Reference 16)

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13. ABSTRACT

A computer program has been written to calculate the heat transfer on the windward side of a delta wing in hypersonic flow. The program utilizes the small crossflow theory of Vaglio-Laurin. Heat transfer values are calculated for both the laminar and turbulent boundary layer cases, and results are compared with experimental data on delta wings available in the literature.

14 KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Three-Dimensional						
Hypersonic Flow						
Delta Wing						
Heat Transfer						